James W. Mar
P.O. Box 51281
Pacific Grove, CA 993950

12 December 1993

Dr. John M. Deutch
Under Secretary of Defense for Acquisitions
3010 Defense Pentagon
Room 3E933 Pentagon
Washington D.C. 20301-3010

Dear Dr. Deutch,

Attached are the *Findings and Conclusions* and the *Recommendations* of the C-17 Executive Independent Review Team. Additionally, there are *Appendices* that provide definitions and background information.

Sincerely yours,

James W. Mar

James W. Mar
Chairman EIRT

#767

FINDINGS & CONCLUSIONS

0. The EIRT received excellent and exemplary support from both the SPO and McDonnell Douglas Aerospace engineers. Their competence, enthusiasm and dedication were very evident during our discussions. We greatly appreciate their patience in answering our questions. In order to meet the very demanding USAF requirements, Douglas has had to use the best of current engineering philosophy, processes and analytical methodologies. Compared to other transport category aircraft, the C-17 wing structure is efficiently designed and the EIRT believes Douglas engineering deserves praise for their accomplishment.

1. The static test of the C-17 wing during the loading condition designated as SP 5005M (maximum wing up-bending) was terminated by a failure located in bay 26 of the upper surface of the left hand wing at 144% of test limit load on September 10, 1993. Damage to the left hand wing was extensive but most of this was secondary and can be attributed to the paroxysm of the loading jacks after the initial failure.

2. Test SP5005M verified the strength of bay 26 to its specification level of 150% of design limit load even though the test did not reach its goal of 150% of test limit load. Due to the manner in which the test loads simulate the design limit loads, 150% of design limit imposes the same stresses as does 144% of test limit in bay 26 because the test loads do not account for the relieving effect of the chordwise bending caused by the winglet.

3. The EIRT believes the entire upper surface of the wing has been qualified through a judicious combination of tests and analyses. 79% of the upper surface of the wing has been fully qualified by tests to 150% of test limit or design limit by SP5030M and SP5005M even though SP5005M did not reach its objective of 150% of test limit load. Of the remaining 21% of the upper surface, some portion is within one per cent of the stresses sufficient for qualification, another portion is within two percent and the largest insufficiency is 4%. The engineering methodology to predict that a failure will not occur has been verified by the more than 600 separate beam-column structural components throughout the wing that did not fail under the loadings imposed by test SP5030M which attained 150% of test limit and by test SP5005M which attained 144%. Consequently, the EIRT believes it is appropriate to use analysis for the extrapolation of the experimental data from 144% to 150%. Therefore analysis is the means whereby the remaining 21% of the upper surface is qualified. Historically, this is the approach that has been used.

4. The findings cited in (2) & (3) support the EIRT conclusion that a rerun of condition SP5005M to 150% will not yield significant additional information on the strength of the wing.

5. One of the abnormal conditions which increases the design bending moments on the wing is the so-called fuel imbalance that accounts for a potential system failure in the fuel system. This fuel imbalance is set at 8000 pounds for the C-17. (It was learned that the fuel unbalance for the MD-11 freighter is 2500 pounds.) A reduction in the design bending moments accrues for the wing if the fuel imbalance can be reduced to 3000 pounds. Analyses and tests on other aircraft have demonstrated that this is readily achievable. The flight tests revealed an additional reduction in bending moment due to aileron effectivity and the fact that the flight control system is set so that the ailerons are not symmetrically deployed to counter the imbalance. Both of these changes in the specifications would result in tests SP5030M and SP5005M having qualified 95% of the upper cover of the wing.

6. The failure at 144% of limit condition SP 5005M was initiated by the crippling of the front spar cap inboard flange in bay 26 of the left hand wing. The causes of the premature failure are as follows:
   a. The spar web in bay 26 had been weakened by the prior tests of the wing. This weakened web induced additional loads that over-stressed the spar cap. Strain gage response reveals that prior tests had caused irreversible damage to the shear carrying capability of the web .
   b. The flange of the spar cap was undersized due to a machining error. This means the strength of the flange, i.e., the crippling allowable, is reduced below the value used in the design.

   The right hand wing did not exhibit any of these same symptoms. The EIRT concludes this failure is peculiar to bay 26 of the left hand wing of the static test article; there are no implications for any other regions of the wing and hence modifications to the wing are not required.

7. The EIRT endorses the so-called *productionized* wing and believes the use of larger aluminum extrusions is fundamentally superior to the use of steel doublers. The EIRT accepts the structural qualification by analysis of the larger extrusions. In this regard, the steel doubler reinforced members have been qualified by tests SP5030M and SP5005M. This gives assurance that analysis can be used because the structural parameters of the larger extrusions are within those of the steel reinforced z-stringers. From a fatigue standpoint, the use of the extrusions is inherently better because the thousands of fasteners used in the steel doubler members have been eliminated. Additionally, the upper surface of the wing has ample margin to meet the durability requirements and the effect on the lower surface is beneficial but rather insignificant.

8. The EIRT endorses, in principle, the steps being taken to modify the static test article so that the remaining static tests can be accomplished. In order to apply the loads to the right wing, a large part of the left wing has been removed.

The remaining stub of the left wing is being reinforced and modified to apply the loads necessary to balance the test loads on the right wing.

9. There are a number of major tests to be accomplished before the airframe of the C-17 is fully qualified. The requirement is that all of these tests be carried to test ultimate. The portions of the airframe to be exercised in the remaining tests include downbending of the wing, bending of the fuselage, fuselage frames, the fuselage pressure shell and the pylons. It should be noted that there are locations where the calculated margins of safety are below 5% and hence the possibility of a failure is not remote. In this regard the EIRT is of the opinion that three of these requirements are unrealistically too severe. The dynamic taxi condition, TO93W1, places all of the 159200 pounds of load as fuel in the wing with zero payload. This of course will maximize the stresses in the wing during high speed taxi. There is a fuselage jacking condition wherein there is maximum fuel in the wing while the airplane is completely supported on jacks. This is a *back-breaker* requirement for the fuselage. Finally the so-called 2P condition requires the testing of the fuselage to the maximum design pressure of 16.6 psi. Each airplane fuselage will be tested, prior to delivery, to the proof pressure of 11 psi which is much larger than the normal maximum cabin pressure of 7.8 psi. This proof pressure test fulfills the specification requirement for qualification of the fuselage structure.

10. The planned sequence of the C-17 wing static test program carried each condition to ultimate before proceeding to the next condition. In retrospect, it was not the sequence to maximize the amount of data acquired with the minimum amount of disruption to the program. A better sequence would have been to carry each condition to limit and then to carry only one composite condition to ultimate. The design deficiency revealed by the test failure of October 1, 1992 would still have been found and neither the schedule disruption caused by the October 1, 1992 failure nor the test failure of September 10, 1993 would have occurred had this latter sequence been in effect. It is even within reason to speculate that the strain gage data from all of the tests to limit load may have revealed the design deficiency, permitting the design changes to be incorporated prior to the conduct of the ultimate test.

RECOMMENDATIONS

1. *The EIRT recommends that each of the remaining full scale static test conditions that involve the wing and fuselage be carried just to test limit load before any condition is carried to test ultimate.* The likelihood of an unexpected failure increases as the load increases above test limit. As has been demonstrated, a failure can inflict serious damage to the wing and/or fuselage, causing at best, a major delay in the schedule or, at worst, the complete loss of the static test article. Such a sequence of completing firstly all of the conditions to test limit will insure that there will be, at the least, data on the limit load response from all of the structure to be qualified by the remaining tests.

2. The EIRT, if asked, would approve a reduction in the number of tests remaining to be carried to test ultimate conditions. This test article has experienced the trauma of two failures, the wing has been loaded six times above limit conditions, seven other times to limit conditions and the left wing now suffers the ignominy of being part of the test fixture. *The EIRT recommends the C-17 Program Office determine which of the presently planned ultimate tests will reveal understanding of the structural behavior not already confirmed by tests to limit and prior tests to ultimate.* In this regard, for example, the beam-column methodology has been validated by the previous upbending tests. The EIRT believes a reduction in the number of ultimate tests can be justified, and there will be concomitant cost and schedule benefits.

3. The EIRT believes that two of the remaining test conditions are unrealistically overly conservative. These are as follows:
   a. The dynamic taxi condition, TO93W1, to meet the airplane gross weight of 434,500 pounds puts 159,200 pounds of fuel in the wing and no payload in the fuselage. This distribution of the 159,200 pounds does maximize the downbending of the wing but it appears to the EIRT wholly unrepresentative of actual operations.
   b. The fuselage jacking condition, JK216M, has the wing completely full of fuel with the airplane completely supported on jacks. This ground condition maximizes the downbending of the fuselage appreciably beyond any flight condition. It does not seem reasonable to the EIRT to jack up the airplane with the wings full of fuel. For example, wheel changes can be accomplished without this kind of jacking.

   Although the structure has been designed to meet these extreme dynamic taxi and jacking conditions, *the EIRT recommends that the severity of these requirements be challenged on the bases of need and rationality.* If relief will be granted, *the EIRT recommends the test loadings be lowered accordingly.*

4. The ultimate design pressure condition, 2p, requires the pressurization of the fuselage to 16.6 psi, Every production fuselage will be proof pressure tested in the factory to 11 psi which is larger than the normal operating pressure of 7.8 psi. The EIRT believes the proof pressure test obviates the need for the 2p test. It should be noted that a fuselage pressurized to 16.6 psi has sufficient stored energy to wreak much damage to the surroundings should a failure occur. Such a test to ultimate design pressure is not required of civil transport category aircraft. *The EIRT recommends that the 2p condition be eliminated from the full scale static test program.*

5. *The EIRT recommends that steps be taken to reduce the allowable fuel unbalance from 8000 pounds to 3000 pounds.* There are issues regarding the C-17 fuel measuring system that need to be resolved, but these appear to be surmountable. Reductions in loads should always be pursued especially if, as is the case for the fuel imbalance, there are no operational implications.

6. *The EIRT recommends that the Program Office continue to challenge the Air Mobility Command on the magnitude of the payload asymmetry.* The EIRT recognizes the load master loses some flexibility in the placement of the payload if the payload asymmetry is reduced. A few tests have been conducted at Fort Hood that give promise of a reduction in the payload asymmetry that is acceptable to the load master.

7. The EIRT was surprised to learn that the C-17 does not have a load alleviation system. For example, such a system was incorporated into the C5-A. A load alleviation system in the C17 would have appreciably reduced structural weight that could have been used to increase range and/or payload. There is an understandable aversion to re-writing the flight control software necessary to achieve load alleviation. As already stated, reductions in loads should always be pursued, even for the C-17 airplane which is meeting all of its structural integrity requirements. A load alleviation system will make it easier to meet future changes in operational usage. *The EIRT recommends that a load alleviation system be developed for the C-17.*

8. In retrospect the USAF requirement to test the C-17 static article to ultimate conditions several times was ill-devised. To meet the ultimate conditions, an efficient structure will be operating in a regime where both material and geometric non-linearities are important. This means the imposition of an ultimate condition will cause some permanent damage even if there has been no obvious failure. Thus, the next test condition is imposed not on the pristine structure upon which the original design analyses were carried out but upon a slightly damaged structure. *The EIRT recommends the USAF re-examine the ultimate test requirements to determine if it is still sensible.*

APPENDIX C DEFINITIONS & NOMENCLATURE

Limit loads are the maximum loads that the aircraft is expected to actually experience during its life. The structure shall exhibit no detrimental permanent deformations after the application of limit loads.

Ultimate loads are limit loads multiplied by the factor of safety which is specified to be 1.5. The structure shall support without failure the ultimate loads. *(Note: Ultimate loads are fictitious because the aeroelastic characteristics of the airplane makes it physcially impossible to develop these ultimate loads in flight. This is in contradistinction to the limit loads which can be developed by the airplane.)*

Design limit loads are the loads used for the design of the C-17.

Test limit loads are the loads applied to the static test article that simulate the design limit loads. The simulation is not exact because of the limitations of the test fixtures.

Test ultimate loads are test limit loads multiplied by the factor of safety of 1.5.

$$\text{Margin of Safety} = \frac{\text{Allowable} - \text{Requirement}}{\text{Requirement}}$$

where the allowable is the capability of the structure and the requirement is what is imposed on the structure by a design condition. For example, the allowable for a member in tension made of the aluminum alloy, 7150, is the tensile ultimate of 85 ksi. If the stress induced by a particular ultimate loading condition is 80 ksi, then the Margin of Safety, M.S., is

$$\text{M.S.} = \frac{85 - 80}{80} = .063 \text{ or } 6.3\%$$

For a structure subjected to multiaxial loadings, such as is the case for the upper surface of the C-17 wing, the determination of the margin of safety becomes much more involved than is shown by this example.

SP5005M : The designation for the loading condition that terminated at 144% of test limit load. SP refers to steady pitch (the aircraft is subjected to a pilot induced maneuver) and the M means modified to the 585,000 pound TOGW.

DG5001: The designation for the loading condition that caused the wing failure on October 1, 1992. DG refers to discrete gust (the airplane encounters a gust) and the M means modified to the 585,000 pound TOGW.

APPENDIX D THE WING STRUCTURE

The wing in the region of the failure is comprised of two spars consisting of spar caps which are mechanically fastened to the spar webs. The upper and lower surfaces of the wing consists of z shaped stiffeners mechanically fastened to the wing skins. Wing up-bending causes the upper surface to be in compression which means that structural instability is the dominant mode of failure. Each z stiffener plus a width of the skin is modeled as a structural entity called a beam-column that is subjected to compression, local bending and shear along the edges of the width of skin. The compression load is caused by up-bending (transverse bending) of the wing; the local bending is due to the combination of airload pressure, crushing pressure induced by wing up-bending, fuel pressure and column compression combined by a beam-column analysis; and the shear is caused by the torsion which accompanies up-bending of the wing. The spar caps plus a width of skin are modeled as structural entities subjected to compression, local bending and shear along the edges of the width of skin. However, column buckling is not critical because the skin prevents column buckling in one direction and the shear web prevents buckling in the perpendicular direction. The length of each beam-column is the distance between wing ribs.

Column buckling determines the allowable for compression, crippling determines the allowable for local bending and the material shear stress is the allowable for the shear. Implicit in the modeling of the upper surface of the wing as a collection of beam-columns is that crippling of each of the components of the z stiffeners or spar caps will not occur at the loads used in the analyses of the beam-columns, i.e., the crippling allowable of each component is above the beam column allowable.

The ability of each beam-column to carry a given load is measured by a factor called the margin of safety, M.S. A zero margin of safety implies that the structure is perfectly designed and does not have any excess weight. There are three components in the calculation of the M.S. for the beam-column, one weighing the effect of column buckling, one weighing the effect of the local bending and one weighing the effect of shear. These three components are combined in an empirical interaction formula that was slightly modified by Douglas after the first static test failure.

The wing of the C-17 is very efficiently designed. This is evident when the weight of the wing is compared to other airplanes. In the region of the failure,

zero margins of safety are shown for the front spar and for the adjacent two stiffeners.

APPENDIX E STRUCTURAL INTEGRITY

The static test failure of 1 October 1992 and the recent failure of 10 September 1993 has raised concerns about the structural integrity of the C-17. It is important to realize that the two tests that caused failure represent only a small part of the total structural integrity program.
A brief review of the major elements in the structural integrity program is herein presented as the background for the findings, conclusions and recommendations made in the main body of this report.

The aircraft structure must possess three main attributes:

1. Strength to sustain flight, landing and taxi loads.
2. Rigidity to prevent detrimental aeroelastic effects such as flutter and aileron reversal.
3. Longevity to sustain the cyclic application of flight, landing and taxi loads for a specified number of flight hours, landings and calendar years.

At a particular region of the airframe, the physical sizes, e.g., thickness of the skin or the spacing of the stiffeners, may be dictated by any one of the three desired attributes. In the C-17, the lower surface of the wing is sized primarily by longevity and the upper surface primarily by strength to resist buckling. Rigidity considerations are a factor in determining the spanwise location of the engines on the wing.

In metal airplanes the repeated application of flight, landing and taxi loads will lead to the initiation of cracks that, if unrepaired, will grow to a size such that the damage tolerance strength requirements will be compromised. The detection of cracks and their repair can be a major maintenance burden. Thus, of the three attributes, longevity is the one that has the greatest impact on operating costs because continual surveillance is required for the detection of cracks.

There are two facets to the strength attribute. The first is the capability to sustain design limit loads without permanent deformations, i.e., the structure must be reusable after experiencing design limit loads. The second is to sustain design ultimate loads without failure but the structure is not required to be reusable after experiencing design ultimate loads and indeed there is no requirement to be reusable after experiencing loads greater than design limits loads. Design limit loads are the maximum of those which the airplane is expected to experience during its life. In contrast, the design ultimate loads, which are legislated to be numerically equal to the factor of safety times the design limit loads, are fictitious in the sense that the airplane cannot be flown so as to develop design ultimate loads. The factor of safety is 1.5. It is impossible to satisfy the ultimate requirement without first satisfying the limit requirement so that from this viewpoint, the limit condition is the more important of the two.

The design ultimate requirement is intended to take care of unk-unks, unknown-unknowns. One can argue that the range of unk-unks for the C-17, the fifth generation of military transport category aircraft, is much less than for the DC-3. Nonetheless, airplanes designed to the factor of safety of 1.5 have been successful in meeting their strength requirements.

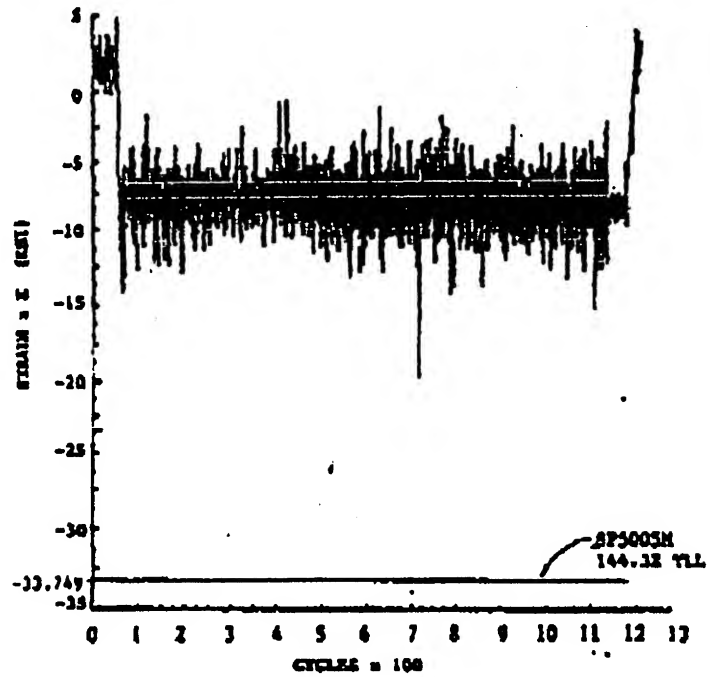
There are three full scale airframes devoted to the verification of the design analyses:

One is a flight test airplane used to verify the aerodynamic loads. This flight test airplane is also used to demonstrate that flutter and other aeroelastic phenomena are beyond the normal flight envelop of the airplane, i.e., to validate the rigidity attributes.

A second airframe is used to verify the longevity attributes by the cyclic application of the loads to be experienced in flight. This test is referred to as the durability test (also known as the fatigue test) by the USAF and its objective is to validate the crack initiation and crack growth characteristics of the airframe. The C-17 durability airframe will cyclic loaded to the equivalent of two lifetimes of usage; presently the durability airframe is at the equivalent of one-half of a lifetime. There are some who would like to see the application of a third lifetime on the durability airframe because recent concerns about aging aircraft, military and civil, are best addressed by a third lifetime of durability testing.

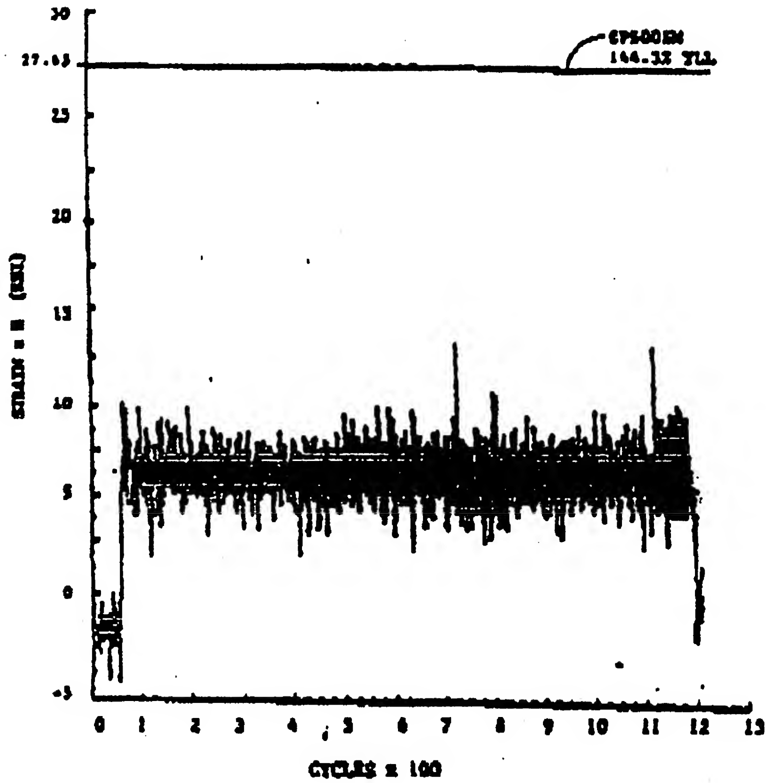
The third airframe is used to verify the strength attributes of the C-17. To date 24 separate tests have been completed on the static test article and 6 more remain to be completed.

The stresses used for the static strength design of the airframe are much larger than those used for the longevity, i.e., durability, design. This is shown in the accompanying traces of strain response from the durability test article, one from the upper surface and one from the lower surface. Both of these traces are taken from the worst case flight in the durability spectrum of loading.



STRESSES IN UPPER SURFACE IN BAY 26

In the upper surface the maximum compressive stress is seen to be approximately 20 ksi in the durability article whereas it reached a shade under 34 ksi in the SP5005M test.



STRESSES IN LOWER SURFACE BAY 26

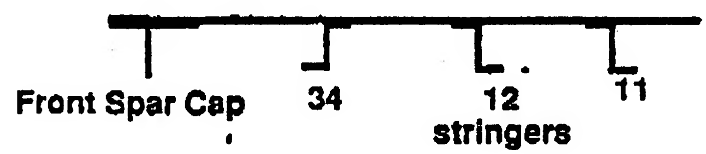
In the lower surface, the maximum tensile stress is seen to be approximately 12.5 ksi in the durability article whereas this same location reached almost 27.5 ksi in the SP5005M test.

APPENDIX F THE LOADING OF THE STATIC TEST ARTICLE

The structural design of the wing utilizes aerodynamic loads caused by a continuous distribution of pressures on the surfaces of the wing. The aerodynamic loading together with the inertia loads caused by acceleration causes a transverse bending (up-bending) moment, a chordwise bending moment, a torsional moment, a transverse shear and a chordwise shear at each cross section of the wing. The chordwise bending moment and chrodwise shear are caused by the drag on the wing and loads on the winglet. These moments and shears are used to determine the sizes of the stiffeners, spar caps and skins. It is important to remember that all five structural effects, (i) transverse bending, (ii) chordwise bending, (iii) torsion, (iv) transverse shear and (v) chordwise shear are included in the design analyses and that the calculation of these effects are based on a continuous distribution of loads.

The test loads that simulate the design loads are applied to the static test article through tension pads bonded onto the upper surface of the wing. Attached to these tension pads are mechanical linkages that are attached by a whiffle-tree assemblage to a system of hydraulic jacks. This loading system does not simulate the effects of chordwise bending nor chordwise shear because it is not practical to place another whiffle tree system onto the wing due to the large deflections experienced by the wing under load. In particular, the chordwise bending caused by the winglet loads are neglected. These chordwise effects which are included in the design of the wing are generally an order of magnitude smaller than thwas overlaoded re knowingly omitted from the static test.

Thus, at a given location in the wing, the margin of safety at 150% of design limit load will not be exactly the same as the margin of safety at 150% of test limit load because the test loads do not exactly duplicate the design loads. For example, the following table compares design and test margins of safety in bay 26 for the spar cap and three adjacent stifferners:



	Front Spar Cap	Stringer 34	Stringer 12	Stringer 11
Margins of Safety at 150% of Design Limit Load	0.04	0	0	0.10
Margins of Safety at 150% of Test Limit Load	0	0	0	0.02

These are the margins of safety, calculated prior to the conduct of the test, for condition SP5005M that caused the failure. The zero margin of safety shown for the front spar cap indicates that the front spar cap is loaded more severely by the test than it should have been. In other words, the test loads cause larger stresses in the spar cap than were used for the sizing of the spar cap.